SMALL UPPER STAGE

AN ORBIT ENABLING HYDRAZINE PROPULSION STAGE FOR THE SMALL SATELLITE COMMUNITY

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ABSTRACT

Under funding from the Naval Research Laboratory (NRL), Rocket Research Company (RRC) has recently completed a one-year study and design effort on the Small Upper Stage (SUS). The goal of this effort was to create a generic, low-cost propulsion stage design capable of being integrated to several different launch vehicles (LV's) and capable of providing a mission enabling orbit transfer function to a variety of different small satellite missions. In order to accomplish this goal, a launch vehicle survey was conducted to establish the current LV capability and environments with respect to small satellites. In addition, a small satellite survey was conducted to establish the orbital needs of the small satellite community. With this information, the SUS specification was created. Based on this specification, the preliminary design of the SUS was developed. This design is capable of being integrated onto several different LV's and is capable of providing the orbit adjust requirements for more than 75 percent of the small satellite missions identified. The SUS design has capabilities of providing small satellites with several orbit transfer maneuvers from a LEO parking orbit (e.g., Hohmann transfer, circularization, plane change and deorbit). The SUS, containing a single or multiple small satellites, can be integrated onto a host LV as a primary on the SLV's, as a secondary on the MLV's or as multiples on a dedicated LV. With the completion of a proposed SUS development and qualification program, the LV and small satellite communities will have a new asset to enhance current capabilities and enable future missions without the need to fund a nonrecurring program for a custom propulsion stage.

Introduction

Rocket Research Company (RRC) has recently completed the first two tasks of a seven-task SUS design/qualification program. The first task was to create a specification for a small satellite upper stage based on current launch vehicle capability and small satellite orbital needs. The second task was to create the SUS preliminary design based on the specification. Contained herein is a summary of the final report which documents the results of the first two tasks of this program.

Launch Vehicle Survey

To establish the launch vehicle portion of the SUS specification, a launch vehicle survey was conducted. Users guides were obtained for each of the currently available U.S. launch vehicles (Fig. 1). From these guides, the various structural and thermal environments were determined and the SUS requirements were established which encompass all of the published launch vehicle data. This was done such that the SUS could be flown on any of the launch vehicles. Figures 2,

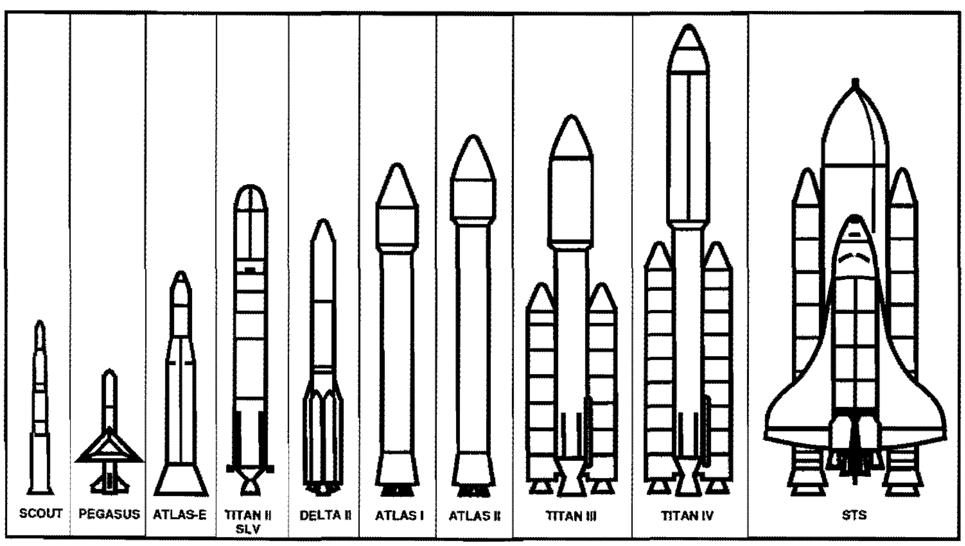


Fig. 1 Existing and near term launch vehicles

3, 4, and 5 show the structural requirements of random vibration, steady-state acceleration, shock and acoustics, respectively. In addition, the maximum radiative heat flux and the free molecular heat flux were determined to be 330 Btu/hr-ft^2 (for 3 minutes) and 360 Btu/hr-ft^2 (decaying to zero in 2 minutes), respectively.

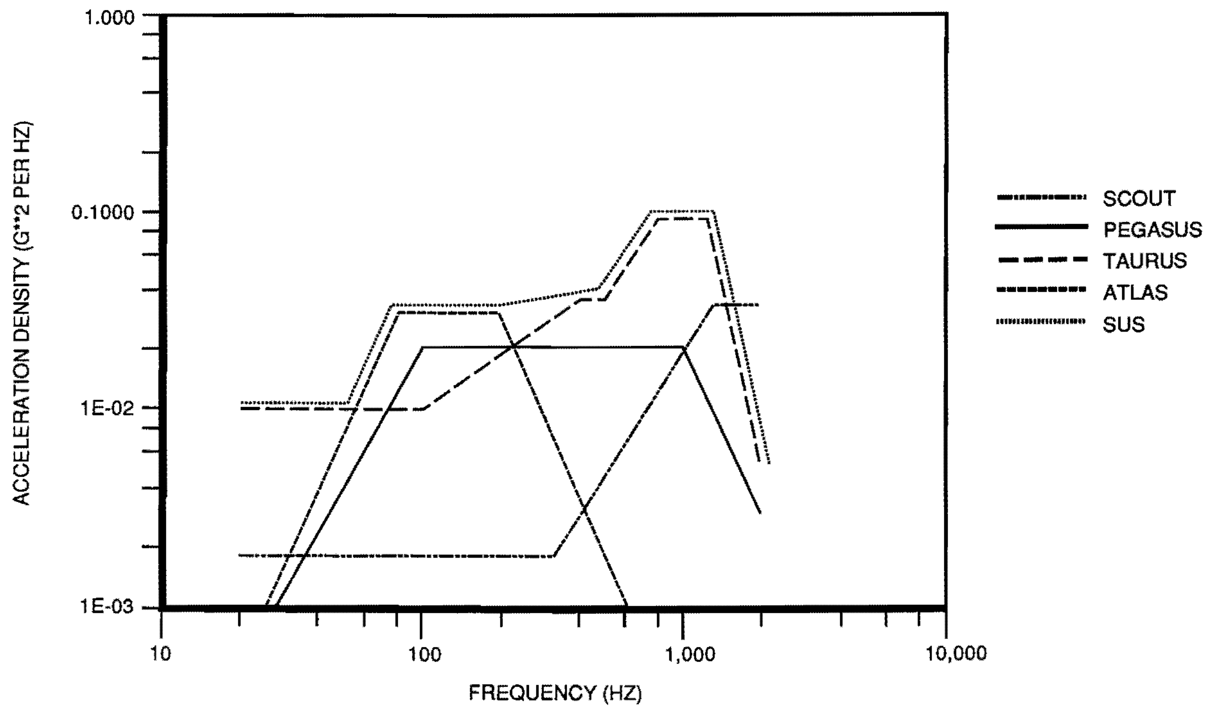


Fig. 2 Launch vehicle flight random vibration

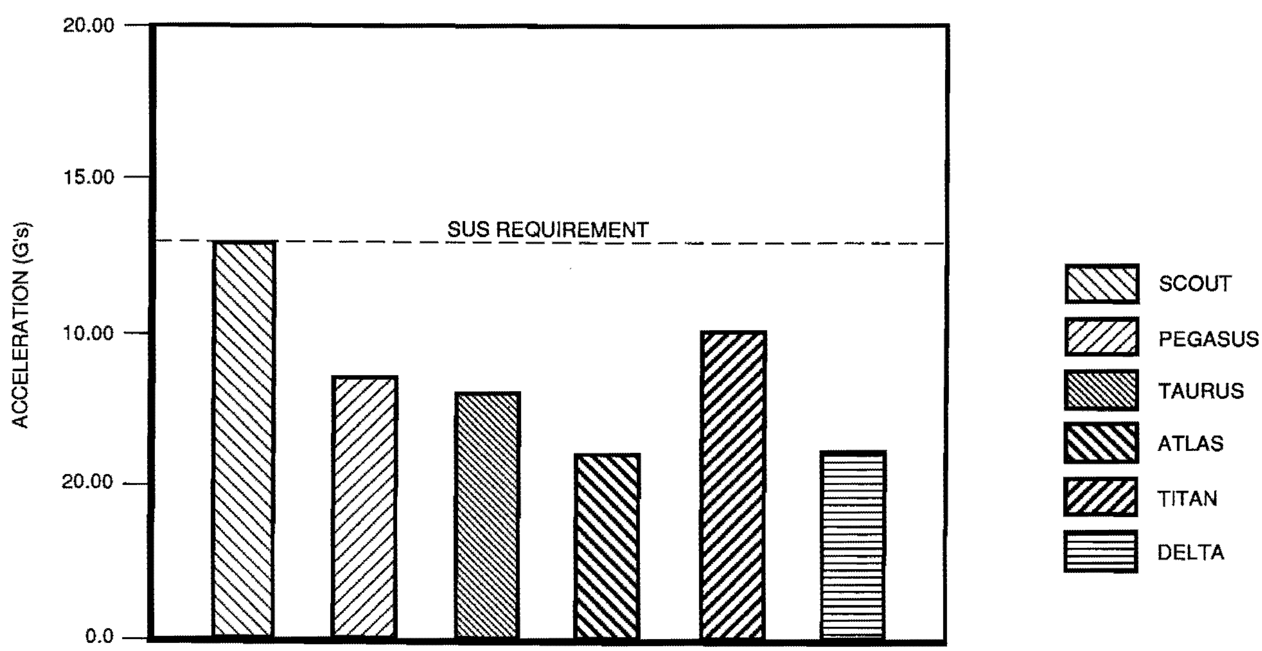


Fig. 3 Launch vehicle acceleration

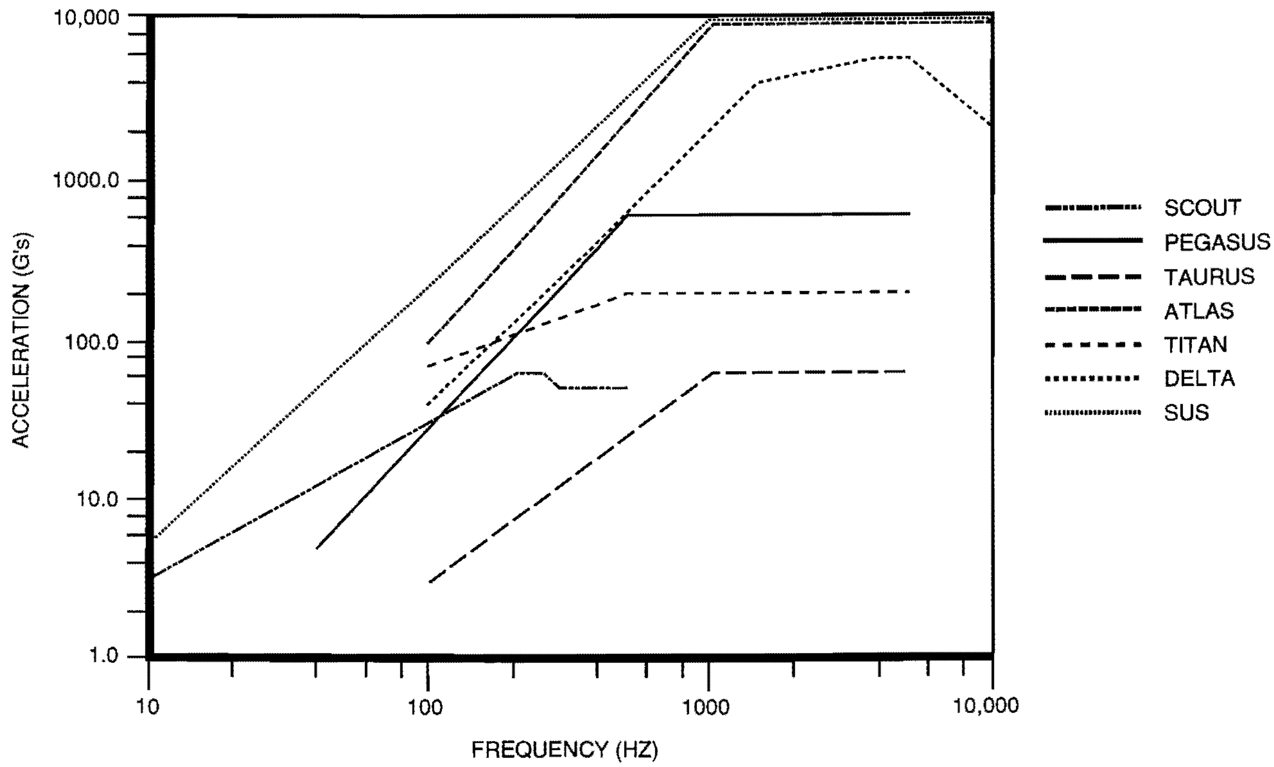


Fig. 4 Launch vehicle shock environment

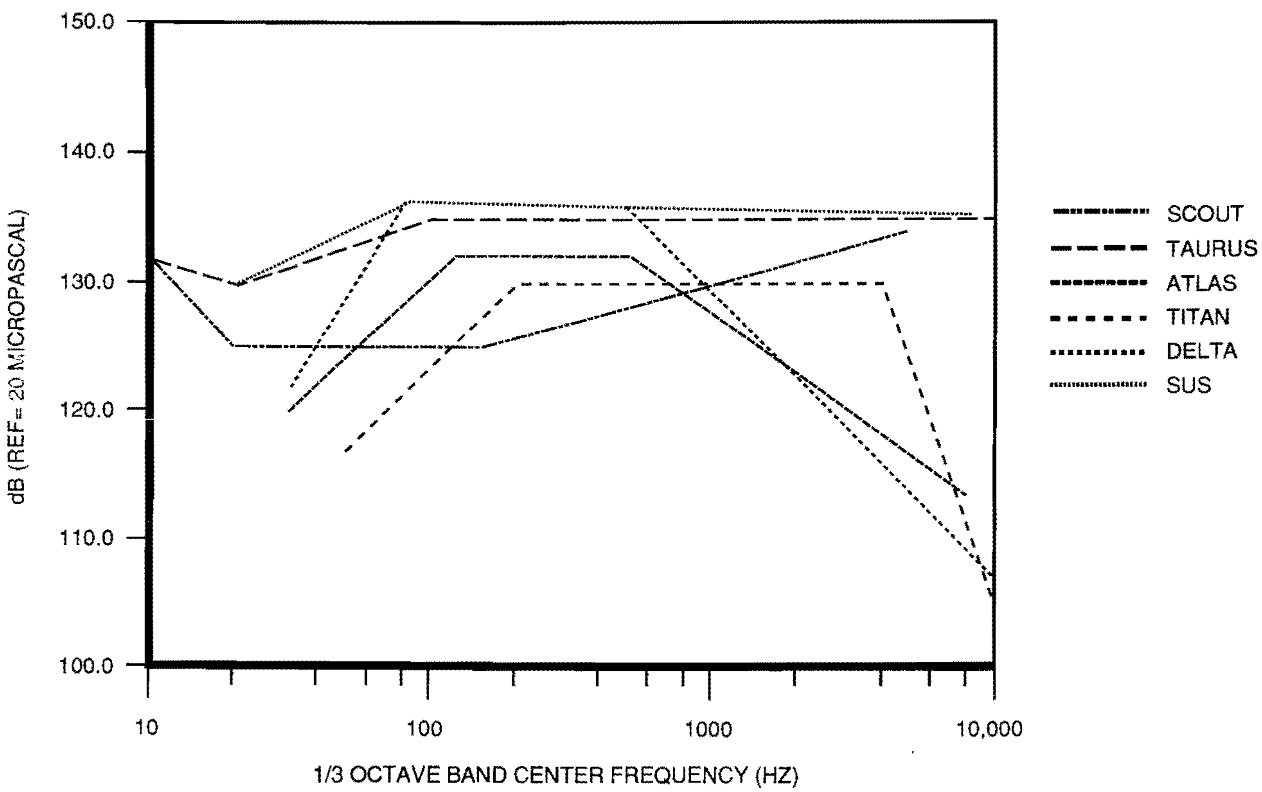


Fig. 5 Launch vehicle sound pressure levels

This launch vehicle survey proved to be valuable on helping determine other SUS requirements which are listed as follows:

- Maximum envelope of 41-in. Dia x 32-in. long – which allows the SUS and a small satellite to fly as a secondary payload on Titan II and Titan IV.
- Starting orbital accuracy of ±5 nm altitude and ±0.15° inclination – which allows for a determination of the SUS accuracy budget.
- The SUS must provide a state vector at LV separation, since the launch vehicles currently do not have available data bus interface to the payload which can transfer this state vector information.

Small Satellite Requirements

To establish the small satellite requirements portion of the SUS specification, a small satellite survey was conducted, a small satellite data base was generated, and appropriate portions of the data base were enveloped with chosen parameters for the SUS specification.

In order to conduct the small satellite survey, the ground rules were established that a small satellite, for the purposes of this survey, weighs less than 1000 lbm, is Earth orbiting and is dedicated to a U.S. mission. The small satellite survey included both small satellite initiatives and launched small satellites. Launched small satellites were included in the data base since the quantity of identified small satellite initiatives was limited and the small satellite history will reinforce the SUS requirements established for the initiatives.

The small satellite initiative survey was conducted by soliciting information from more than 45 small satellite organizations (both users and builders). This established a data base of 35 program initiatives. The data base for launched small satellites was established based on select information from the TRW space log. The launched small satellite data base included all U.S., Earth orbiting, satellites weighing less than 1000 lbm (more than 450 satellites).

With the computerized small satellite data base now established, it is a simple matter to evaluate the parameters needed to establish the SUS requirements. Of primary interest are the parameters of satellite mass and orbit altitude since via the rocket equation, these parameters (along with the SUS dry mass) will define fuel mass. Other parameters such as inclination angle, diameter and length were used for design guidance.

Figure 6 shows a plot of desired orbit altitude as a function of percent of the small satellite initiatives. It is clear from this figure that more than 80 percent of the identified initiatives will be LEO missions. There was one mission in GTO and a few in GEO. Referring to Fig. 7, which shows altitude as a function of percent of small satellite initiatives in LEO (<800 nm), it can be seen that selecting the orbit altitude of 500 nm will envelope all of the LEO small satellite initiatives. Therefore, it seems reasonable that SUS should have the total impulse capabilities to provide a Hohmann transfer from 100 to 500 nm.

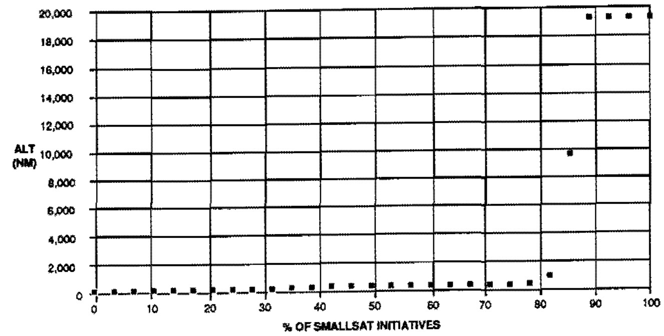


Fig. 6 Altitude vs. % of SMALLSAT initiatives

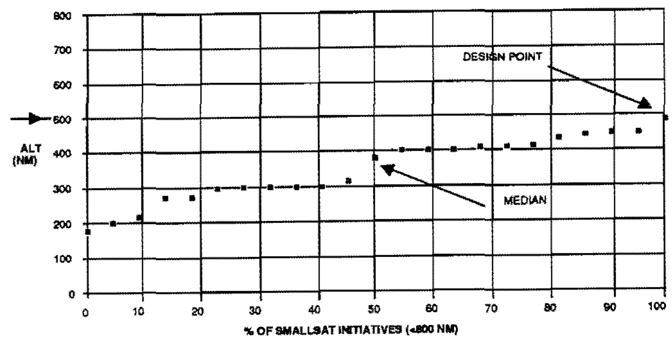


Fig. 7 Altitude (<800NM) vs % of SMALLSAT initiatives

The other parameter of primary interest is small satellite mass. Fig. 8 plots the mass of the small satellite initiatives as a function of the percent of the initiatives. Note that this plot's slope takes on a significant increase above 400 lbm. Therefore, it seems reasonable to chose 400 lbm as the small satellite maximum mass, since this mass envelope is approximately 75 percent of the small satellite initiatives. As in the small satellite initiatives, similar plots were generated for the already launched small satellite data base. A summary of initiative versus launched small satellite median data is shown in Table 1.

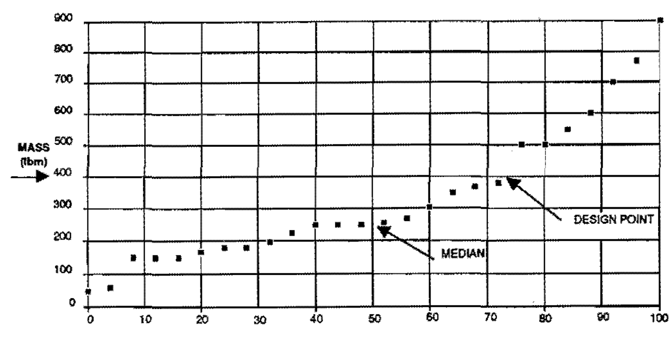


Fig. 8 Mass vs % of SMALLSAT initiatives

Table 1. A typical small satellite using median data

	Initiative	Launched
Altitude*	380 nm	430 nm
Mass**	260 lbm	150 lbm
Inclination**	82°	82°
Diameter**	23 inches	24 inches
Length**	29 inches	27 inches

* Median point for small satellites in LEO (<800 nm).
** Median point for all small satellites.

This data shows the only significant change between the initiatives and the history is in the parameter of mass (which makes sense because of the development of more capable space boosters over the 33-year history of U.S. spaceflight).

Table 2 summarizes the chosen SUS specification parameters of mass and altitude along with the percent of missions incorporated from both the initiatives and the historical data bases.

Table 2. Small satellite-driven SUS requirements

	Amount	Small satellite initiative incorporation	Launched small satellite incorporation
Altitude	500 nm	100% of LEO (<800 nm)	70% of LEO (<800 nm)
Mass	400 lbm	75% of All	78% of All

The minimum total impulse requirement of the SUS is now defined. (Sufficient ΔV capability to perform a Hohmann transfer from 100 nm to 500 nm on a 400 lbm satellite.) With this specification requirement the SUS fuel carrying capabilities can now be defined. Referring to Fig. 9 and selecting the parameters 500 nm along with a 400 lbm satellite, the minimum SUS fuel load is approximately 120 lbm of hydrazine (I_{sp} ~225 sec). Of course, in order to generate this parametric plot of the rocket equation, the SUS dry mass must be known. Therefore, Fig. 9 is the result of an iterative process using the dry mass of the final SUS design (approximately 120 lbm).

As will be noted in the design section later, the resultant SUS configuration has the ability to carry approximately 170 lbm of fuel which is sufficient to carry a 700 lbm satellite from 100 nm to sun synchronous orbits. Structurally, the SUS design still has sufficiently positive margins of safety for satellite masses up to 700 lbm. Therefore, the basic SUS design is much more capable than the minimum specification requirements

The other primary area of interest in the small satellite requirements is final orbit accuracy. The small satellite and user communities had difficulty in defining these parameters. The parameters of apogee ±25 nm, perigee ±25 nm and inclination angle ±0.5° were selected for the SUS specification which appeared to envelope most requirements (±0.5° is arguable for sun synchronous orbits). In fact, the basic SUS design has estimated accuracy capabilities which are approximately an order of magnitude better than the above tolerances.

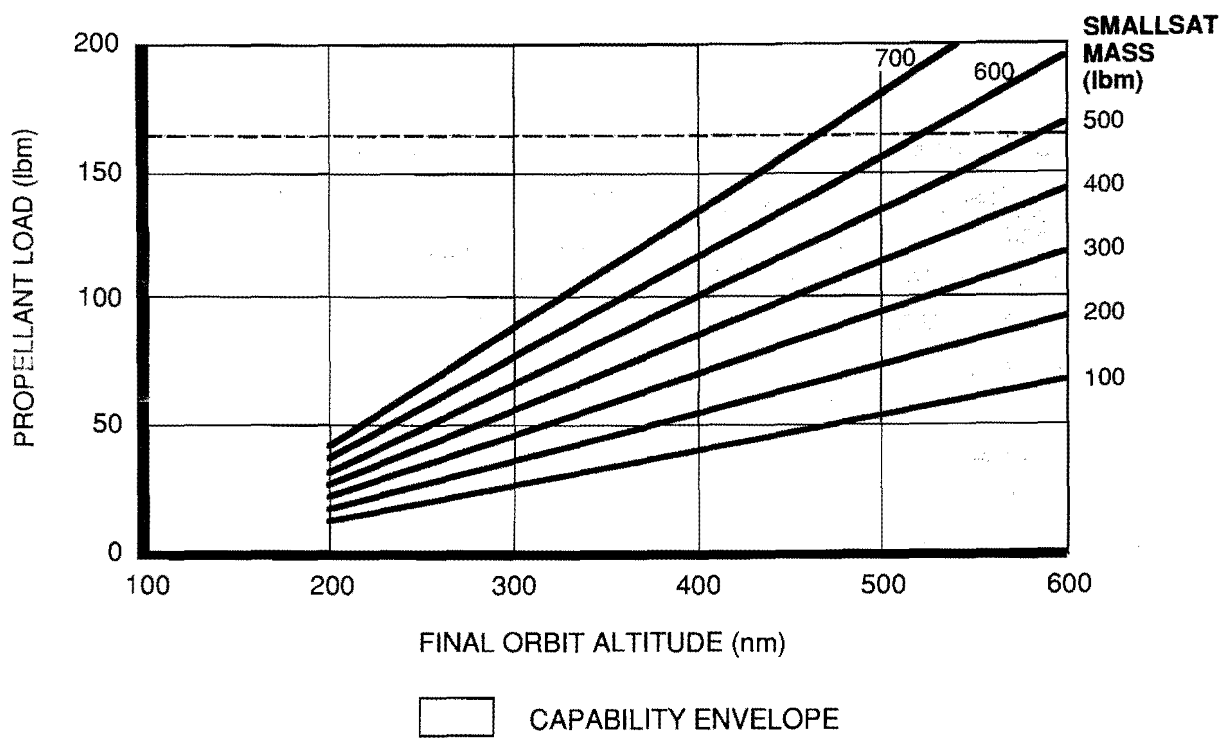


Fig. 9 SUS transfer capability (Initial parking orbit = 100 nm)

SUS Specification

Based on the data obtained in the launch vehicle and small satellite surveys, the SUS prime item development specification (RRC CS-0252) was written in accordance with the specification practices of MIL-STD-490. A summary of survey-driven specification requirements is shown in Table 3.

SUS Design

Cost is of primary importance in the SUS design. To meet aggressive cost goals established for the SUS, particular attention was given to component selection. Component selection was limited to already proven hardware, thereby reducing or eliminating nonrecurring charges and insuring continued supply at predictable prices.

The SUS vehicle is configured as a generic upper stage capable of performing a variety of orbital maneuvers including Hohmann transfer, plane change, and circularization from an elliptical orbit. In addition to mission flexibility SUS also posesses the required subsystem flexibility and computational capability to conform with various satellite requirements and mission profiles which could include complex reorientation maneuvers or various ΔV thruster burn profiles.

The basic SUS design is shown in Fig. 10 and as illustrated in Fig. 11, this design consists of seven distinct subsystems including structure, separation, propulsion,

Table 3. SUS base specification requirements summary

Characteristic	SUS specification requirement
Orbit accuracy	* Apogee and perigee ±20 nm * Inclination ±0.35°
PLF envelope	41-in. diameter, 32-in. length
Thermal environment	
Launch	60 – 80°F until liftoff 330 Btu/hr-ft ^{2}, 3 minutes, from PLF 360 Btu/hr-ft ^{2}, decays to zero in 2 minutes, from FMH
On-Orbit	Maximum flux Solar flux = 450 Btu/hr-ft ^{2} Albedo = 0.32 earth reflectance Earth IR = 80 Btu/hr-ft ^{2} Minimum flux – zero flux from sun and earth
Vibration environment	9.6 g rms
Shock environment	9,000 g's max
Acoustic environment	149.8 dB overall
Acceleration Environment	13 g's any direction
Orbit transfer	Hohmann transfer – 100 to 500 nm
Satellite weight	400 lbm

* Relative to the host vehicle

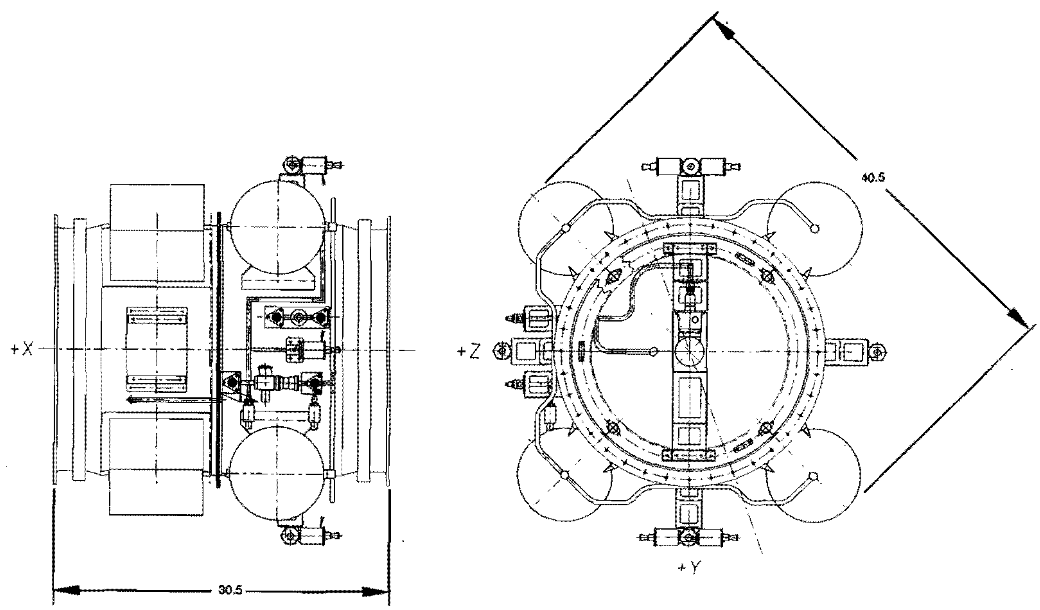


Fig. 10 Baseline SUS design

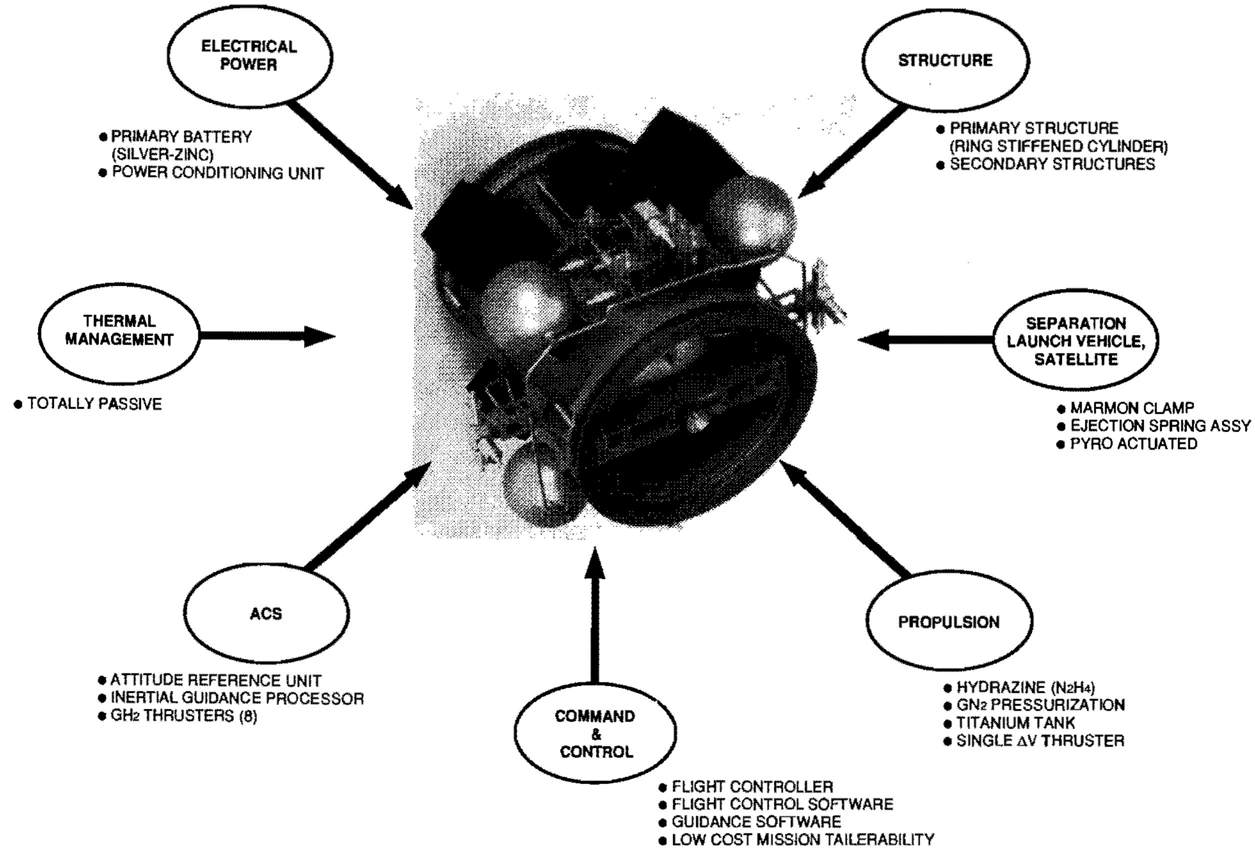


Fig. 11 SUS subsystem overview

command and control, attitude control, thermal management, and electrical power. A brief description of each of these subsystems follows.

Structure

The basic SUS structure, shown in Fig. 12, is a ring-stiffened cylinder inside of which is mounted a 22-in. Dia. hydrazine propellant tank. Attached to the exterior of this structure are various other components which define a diameter of 41 inches and a length of 30 inches.

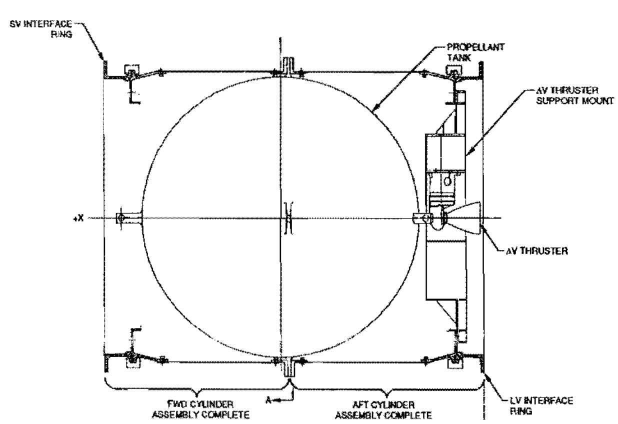


Fig. 12 Structure – Section through primary structure

This structure separates at the tank mounting flanges which provides two convenient substructures. The aft cylinder supports the propulsion hardware, and the forward cylinder supports the avionics such that these sections can be independently assembled and tested in a cost-effective manner.

The SUS structure is capable of supporting a 700-lbm satellite under superimposed random vibration and steady-state acceleration loads from any input axis.

The completed SUS has an inert weight of approximately 120 lbm, and (if loaded to maximum fuel capacity) has a wet weight of approximately 300 lbm.

Separation

The purpose of the separation system is to release the ascent constraint between the SUS and either the launch vehicle or satellite and provide sufficient relative velocity to prevent recontact of the two vehicles. Additionally, the separation system is designed to minimize tipoff and provide for capture of all loose hardware. The SUS separation systems utilize a well established manacle ring clamp (or V-band clamp) configuration with a two-piece clamping ring at each interface (see Fig. 13).

The two halves of each manacle ring clamp ring are attached with tensioning bolts and nuts at two places with each bolt passing through a pyro activated bolt cutter.

Four ejection spring assemblies are located uniformly about the cylinder support ring at each separation interface. The spring assemblies utilize compression springs which are

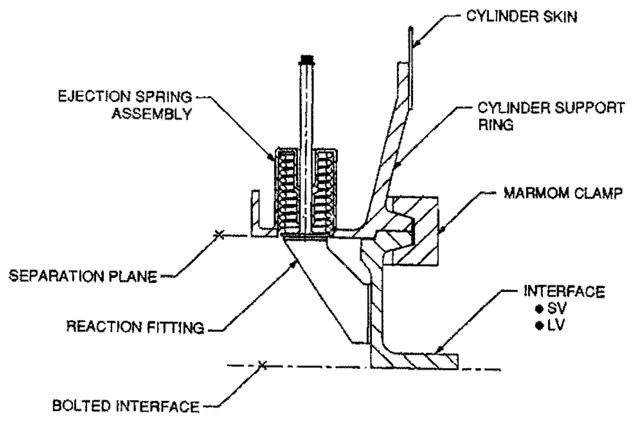


Fig. 13 Section through separation interface

matched for spring rate to develop the required relative velocity (approximately 1.0 ft/sec) between the separating vehicles while introducing minimum tipoff. As a consequence of the pin design selected to introduce tension bolt loads into the manacle ring, the separation system has a redundant separation mode. In the event a bolt cutter failed to activate, the actuation of the remaining bolt cutter with rotation of the manacle ring halves about the pins at the unactuated joint allows for separation of the vehicles.

Propulsion

The propulsion system design philosophy is to minimize cost while building in system flexibility and accommodating range safety regulations. This is accomplished with a simple design using flight-proven components.

The propulsion system schematic, Fig. 14, shows a single ΔV thruster, a single hydrazine propellant tank and a pressurized nitrogen gas storage system. The GN_2 pressurants are serviced by a fill/drain valve and monitored through the use of a pressure transducer. A pyro enable valve seals the pressurization system until activated, at which time GN_2 is fed through a filter to the pressure regulator. The output of the regulator is 365 psi with a minimum regulator inlet pressure of 465 psi. The use of a relief valve at this point provides transient over pressurization relief (with recovery) to prevent ACS thruster over pressurization. The GN_2 pressurant is fed from the regulator to the propellant tank passing through a check valve and pyro enable valve to isolate the pressurization side of the hydrazine tank from the GN_2 system components. For missions with propellant loads of 120 lbm or less, the propulsion system could be operated in a blowdown mode and the pyro valve not operated or the line between removed. The illustrated system is the baseline design because it provides the flexibility to support a variety of propellant loads, up to 170 lbm. A fill/drain valve services the pressurization port of the propellant tank allowing for pre-pressurization to decrease the required GN_2 storage system capacity. A pressure transducer at the pressurant port provides a means for monitoring of the pressurized propellant tank. A 22-inch I.D. spherical flight-qualified propellant tank (Centaur) with an internal volume of 5555 cubic inches provides for propellant storage. The tank baselined has a maximum weight of 14.0 pounds, an operating pressure of 375 psig and with an offset propellant feed port (allowing for reduced overall assembly length) has a 99 percent expulsion efficiency. At the propellant feed port, a fill/drain valve provides for propellant loading, system check-out and detanking, if required. A pyro enable valve in the propellant feed line in series with the propellant valve isolates the hydrazine propellant from the ΔV thruster until actuated. A service valve is located in the propellant feed line between the

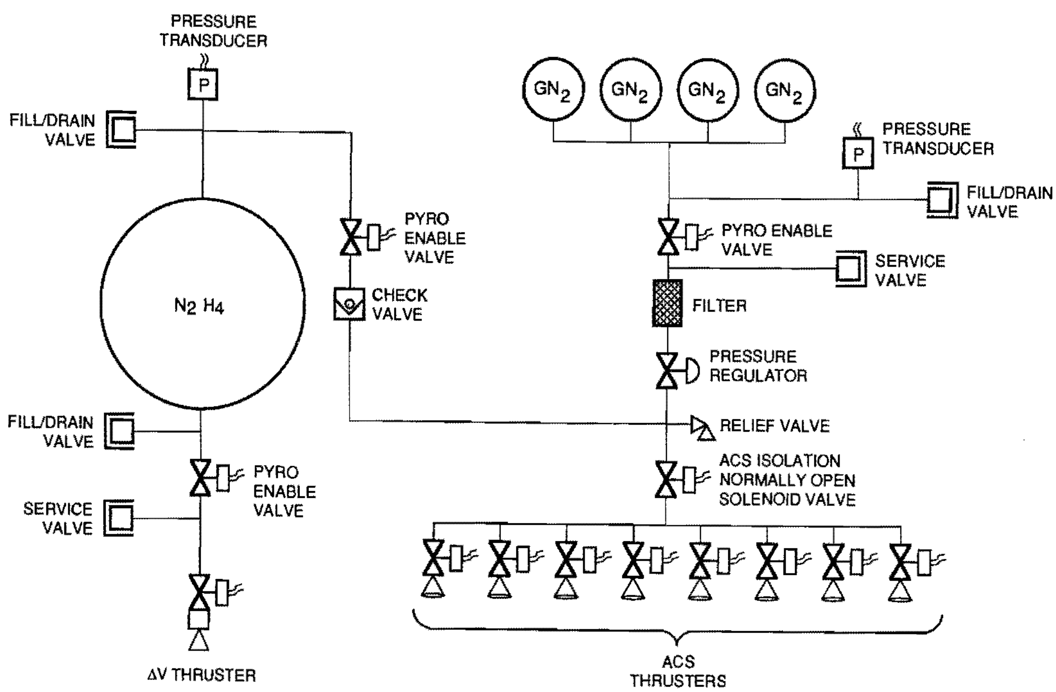


Fig. 14 SUS propulsion system schematic

pyro enable valve and the propellant valve as a test port. The baseline propellant valve is flight qualified and provides a minimal pressure drop of 9.5 psid at the 0.22 lbm/sec hydrazine flow rate required for the 50 pound ΔV thruster. The 50-lbf thruster is the flight qualified RRC Model MR-107 thruster being used on the Atlas II Roll Control Module. This thruster with a nozzle axis oriented 90 degrees to the reactor is compact along the thrust axis and contributes only four inches to the overall length of the SUS while providing the equivalent thrust of engines with twice the length in the thrust axis. Alignment of the thruster is referenced to the SUS's satellite vehicle interface, thereby eliminating major influences of intermediate structure. Alignment adjustments are made by shimming and the use of eccentric spacers at the thruster mounting interface.

Command and Control

The primary responsibility of the Command and Control (C&C) subsystem is administration of mission event sequences. This includes control of all SUS functions from SUS power-up through satellite separation. All event sequencing is based on time deltas referenced to an initial time-zero established by the launch vehicle. The C&C subsystem receives system status information in a variety of formats including launch vehicle command discretes, subsystem monitors such as temperature and pressure transducers, in addition to an IGS interface as illustrated in Fig. 15. This information is processed within the C&C subsystem and command outputs are generated which control various SUS functions. These command functions are responsible for controlling SUS separation from the launch vehicle, propulsion system valve sequencing, satellite separation from the SUS, and IGS calculations. Additionally, the Command and Control subsystem is responsible for all subsystem fault detection and isolation functions.

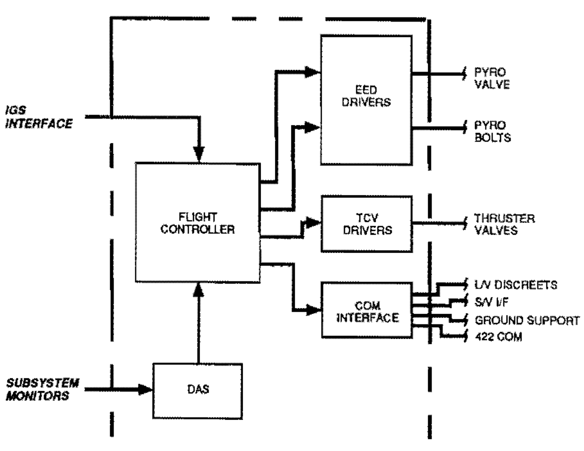


Fig. 15 Electronics control unit 1 (ECU)

Attitude Control

The SUS incorporates a three axis cold gas attitude control system (ACS). This system enables SUS to perform various attitude maneuvers including, large angle reorientation, attitude hold, satellite spinup and orientation, and nutation control functions if required. The attitude control system consists of two major subsystems: the cold gas attitude control thrusters and associated feed system, and the Inertial Guidance System (IGS) sensor electronics.

Eight, cold gas nitrogen attitude control thrusters provide control about the vehicle pitch, yaw, and roll axis. The thruster mounting geometry is such that torques are produce about the pitch and yaw axis, utilizing a single thruster where a pure couple is produced about the roll axis, requiring two thrusters. The roll couple requirement is due to the fact that the roll thrust vector does not lie in a plane that passes through the vehicle CG thereby introducing cross coupling terms in pitch and yaw. It is possible to develop a configuration where this were not the case but the design would be satellite specific reducing the generic capability of the vehicle. Four high pressure GN_2 storage tanks provide the pressurant gas for the ACS thrusters in addition to providing pressurant for the main hydrazine propulsion feed system. A single stage regulator reduces the GN_2 storage pressure (which ranges between 5750 and 465 psia) down to a regulated system pressure of 365 psia. The 16.0-lbm GN_2 capability allows for tip-off rate reduction, multiple reorientations, thrust misalignment, and hydrazine pressurization.

The thruster control commands are generated by the Inertial Guidance System (IGS) which is a subsystem within the attitude control system. A block diagram of this component is presented if Fig. 16. Contained within the IGS are three angular rate gyros which measure motion about the pitch, yaw and roll axis of the vehicle, a microprocessor to correct the rate gyro signals, and an IGS processor which performs guidance computations and generates output commands. The SUS configured with this attitude control system can provide a 100 to 500 nm Hohmann transfer accuracy of ±2 nm for a 400-lbm satellite. The IGS contains a stored state vector for attitude reference throughout the orbit transfer maneuver. Therefore, the SUS mission currently relies on the launch vehicle for controlled pointing at the time of SUS separation.

Thermal

The passive thermal management approach incorporates a selective combination of conductive isolation and radiation surface emittance control. Operational heat input to the vehicle structure is effectively controlled in a manner that does not disturb the overall heat balance of the specified interface. Dissipation of decomposition heat energy (from the ΔV engine) to the vehicle and support structures, thruster valve components and deep space is distributed such that the valve seat, propellant line and injector temperatures are at acceptable levels for all combinations of simultaneous operational duty cycles and environmental conditions. The ΔV thruster can be safely restarted at any time during the specified mission. The temperature range of all electronic components and batteries are favorably maintained without active heater circuits. Passive thermal management techniques combining conductive isolation and radiation emittance are employed to

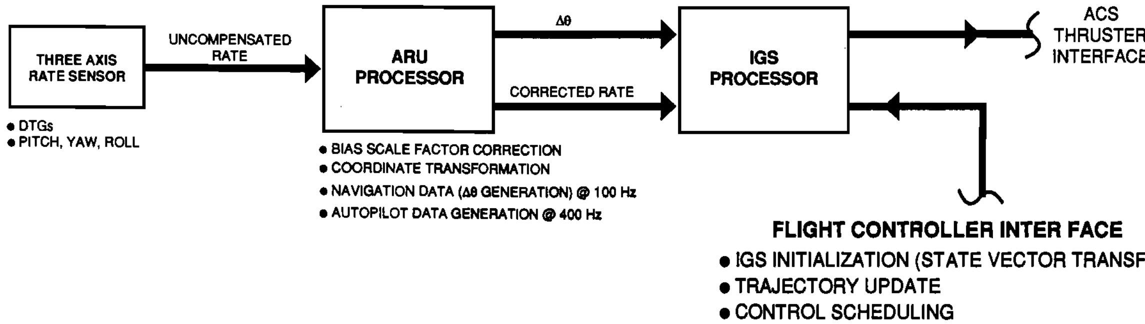


Fig. 16 IGS functional block diagram

maintain the SUS components at acceptable temperature levels when the SUS vehicle is subjected to environmental temperature combinations and mission operation requirements.

Electrical Power

SUS incorporates a flight-qualified Silver-Zinc primary cell. This battery is manually activated with a wet stand time of 150 days. The battery delivers 28 Volts DC and has a capacity of 8 Amp hours. It is packaged in a 7.00 x 3.78 x 4.33 housing and weighs 8.1 lbs.

The Power Conditioning Unit provides regulated DC power to the electronic systems. Referring to Fig. 17, the PCU contains a DC-to-DC switch mode power converter. This converts the 28 vdc battery power to +5, -5, +12 and -12 volt DC power. It also provides protected 28 vdc for Electro Explosive Devices and valve drivers. A bit monitor is furnished for battery monitoring, power converter faults, power on reset conditioning and battery/ground umbilical sensing. The power sequencing and source control circuitry controls the main power source, which can be the battery or ground power. A launch vehicle discrete controls battery power via a pyro switch. All source power is protected and fused. The Power Conditioning Unit is packaged in a 5.5L x 4.75W x 3.65H housing that weighs 3.0 lbs. The unit dissipates 3.2 watts and is over 70% efficient.

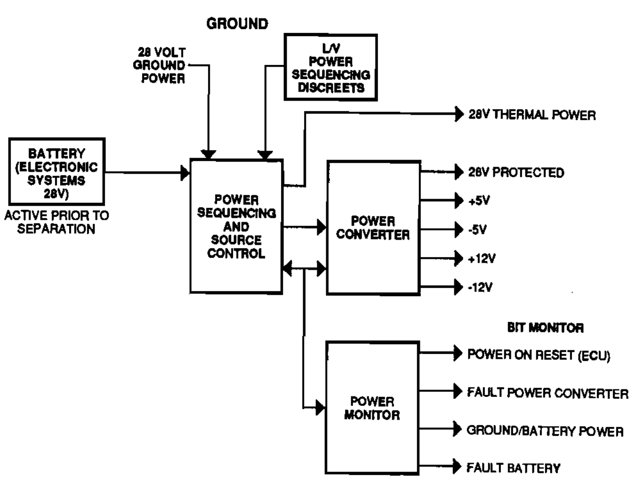


Fig. 17 SUS power control unit/battery

SUS Mission Capabilities

Since SUS is a liquid propulsion stage, it possesses the inherent flexibility of the start and stop capability of the solenoid-controlled ΔV thruster. Therefore, various mission profiles can be programmed into the executive software module in the command and control system. This flexibility, along with the ability to load hydrazine fuel to the desired quantity (up to 170 lbm), affords a very tailorable SUS design which is able to serve a variety of missions. Figure 18 shows the SUS mission capability for a Hohmann transfer, circularization, and a plane change, respectively. Fig. 19 shows a summary sequence of events for a Hohmann transfer of a small satellite riding as a secondary on a medium sized launch vehicle. Fig. 20 shows a medium sized launch vehicle carrying multiple primary satellites, each of which uses SUS to provide a Hohmann transfer. With the proper orbital parameters and sequencing of events, a polar small satellite communications constellation can be quickly established. A corollary to the communication constellation example is a medium sized launch vehicle carrying multiple primaries into an "average" orbit for the multiple missions. The SUS would then be employed to transfer each of the satellites into their proper final orbits.

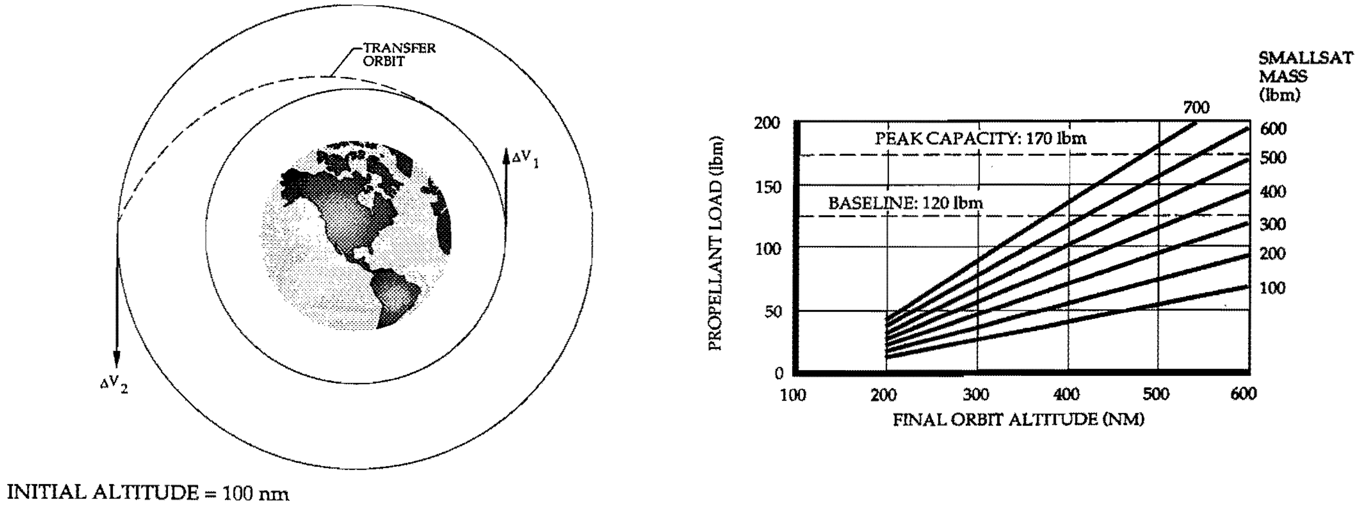
SUS Enhancements

The SUS design represents a very capable, low cost, propulsion stage which can be integrated to a variety of launch vehicles and which satisfies the orbit enabling requirements of the small satellite community. This system has the potential to be enhanced to a LEO small satellite bus by the addition of a payload support module. This module would bolt directly to the basic SUS, and would provide the basic payload support functions of power, attitude control, attitude determination, and telemetry.

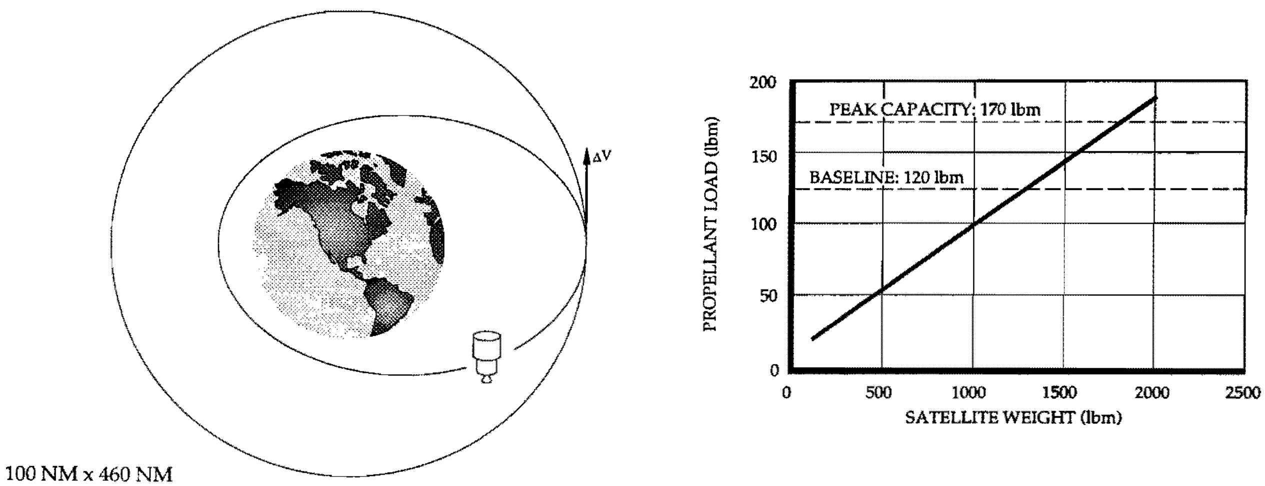
Conclusion

The SUS preliminary design is now complete. This design was established based on current launch vehicle capabilities and the small satellite community's needs. With the SUS generic design, the software and fuel loads are tailorable to enable a variety of different missions in a cost-effective manner. This existence of SUS will eliminate the need for a custom propulsion stage for each future small satellite mission.

HOHMAN TRANSFER



CIRCULARIZATION PERFORMANCE



PLANE CHANGE PERFORMANCE

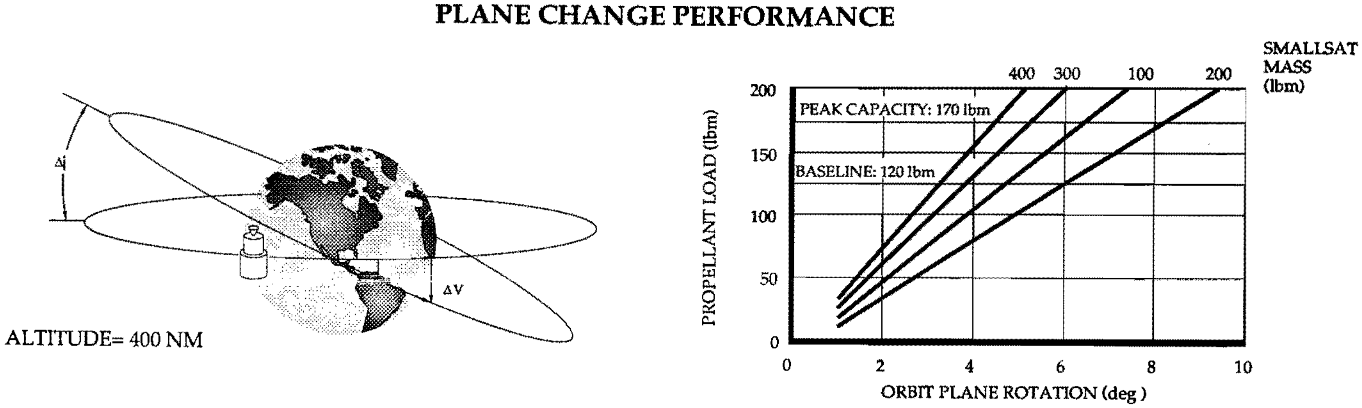
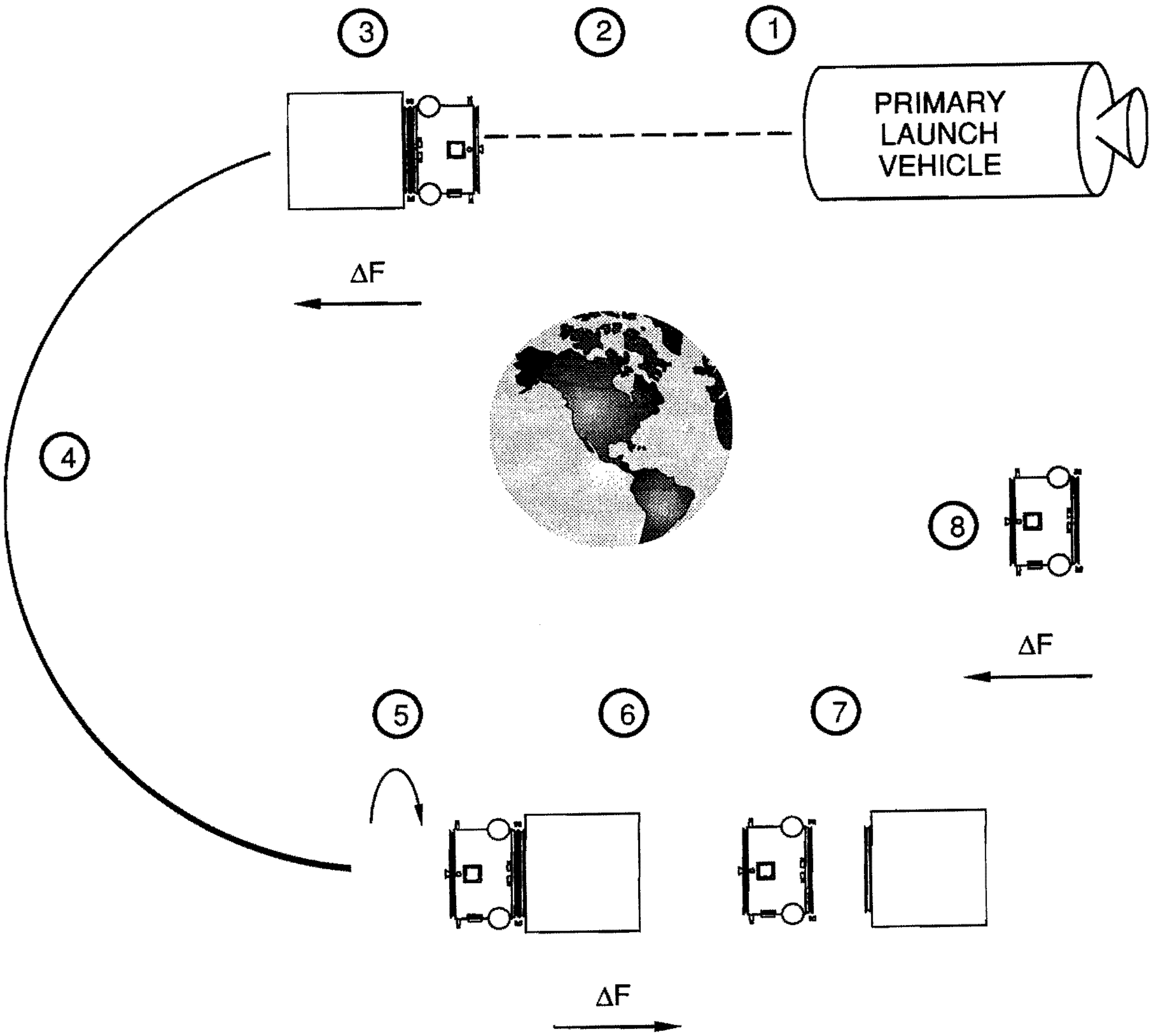


Fig. 18 SUS mission capabilities



EVENT

1. EJECT FROM HOST VEHICLE
2. THREE AXIS CONTROL
3. INSERTION THRUSTER FIRING
4. COAST TO APOGEE,THREE AXIS CONTROL
5. REORIENTATION (180°)
6. CIRCULARIZATION (OR PERIGEE INCREASE) THRUSTER FIRING
7. PAYLOAD DEPLOYMENT (OPTIONAL SPIN-UP)
8. THRUST VEHICLE TO LOWER ORBIT (C/CAM)

Fig 19 SUS Hohmann transfer operation sequence

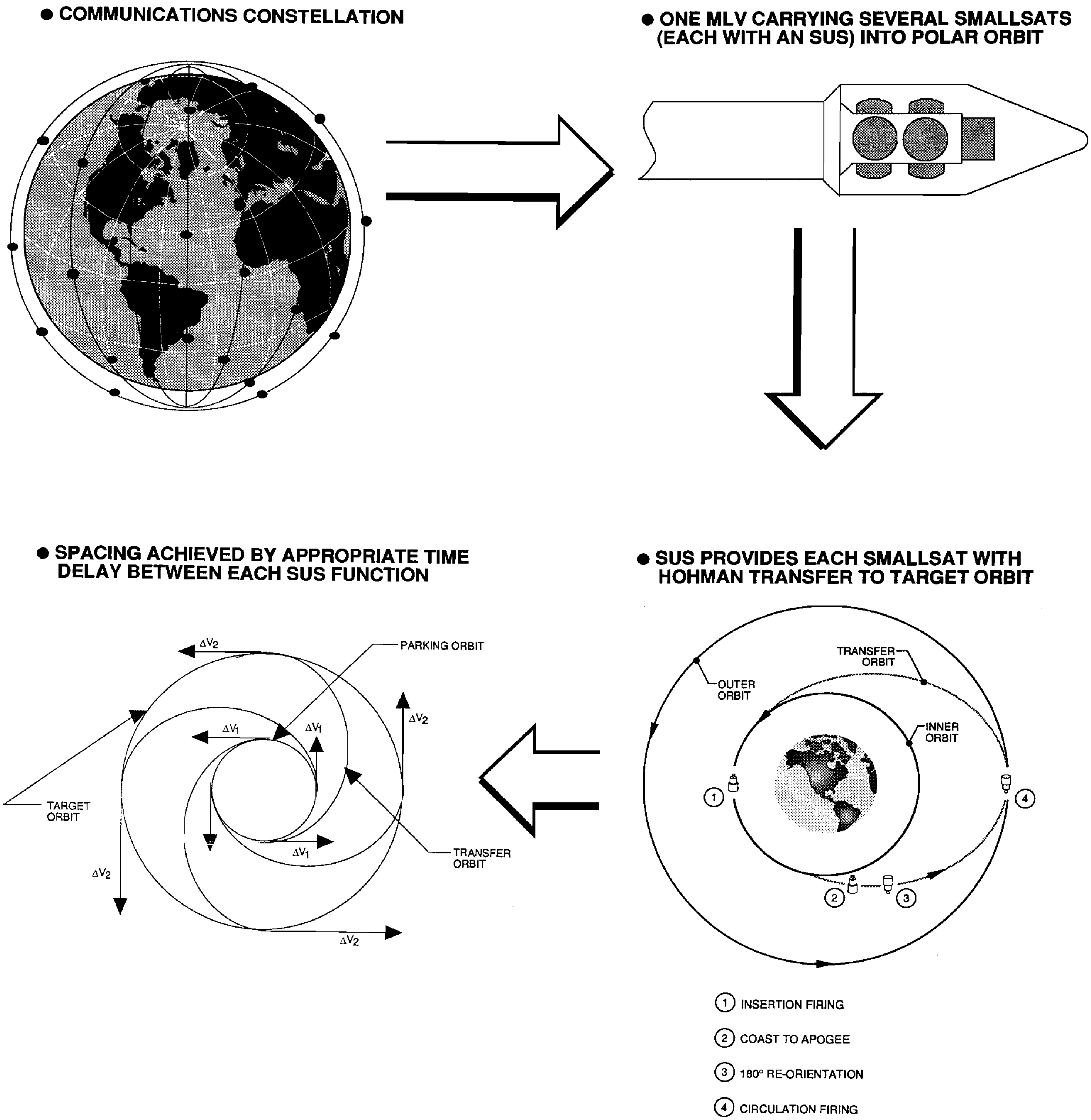


Fig 20 SUS constellation mission example